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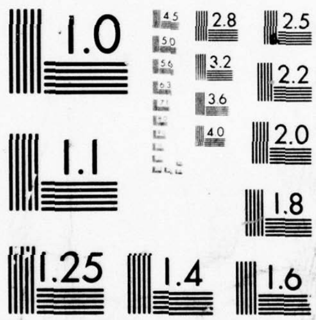

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Technical Memorandum

MAXIMUM LIKELIHOOD IDENTIFICATION OF THE LONGITUDINAL AERODYNAMIC COEFFICIENTS OF THE EA-6B AIRPLANE IN THE CATAPULT LAUNCH CONFIGURATION

Mr. David E. Bischoff
Mr. Roger A. Burton

Strike Aircraft Test Directorate

8 May 1978

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DEPARTMENT OF THE NAVY
NAVAL AIR TEST CENTER

NAVAL AIR TEST CENTER
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MAXIMUM LIKELIHOOD IDENTIFICATION
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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The aerodynamic coefficients describing the longitudinal equations of motion of the EA-6B airplane with landing gear and flaps down were determined from flight test data through use of a maximum likelihood identification algorithm. These identified coefficients were accurately determined and were utilized to both (1) predict short and long period time history responses and (2) duplicate the results of conventional flight test specification requirement test procedures. The identified results provide an example of the type of information available through the application of parameter estimation over

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technology in the areas of airplane data base generation. These data can be utilized for ACLS and Operational Flight Trainer simulations as well as for parametric studies of aerodynamic characteristics.

PREFACE

This Technical Memorandum presents the results of a maximum likelihood identification of the longitudinal aerodynamics of the EA-6B airplane in the catapult launch configuration. The aerodynamic coefficients presented herein were generated for use in a computer simulation of the catapult launch capabilities of the EA-6B airplane.

The flight tests conducted to determine this information were authorized by NAVAIR message 152018Z May 1975 under AIRTASK A510-5102/0535/5W4542-000, Work Unit A53352F-161, of 22 October 1974.

The conventional flight test results conducted in compliance with the above AIRTASK were reported in NAVAIRTESTCEN Technical Report SA-105R-76, Evaluation of Single Engine Catapult Launch Capabilities of the EA-6B Airplane, of 10 February 1977.

APPROVED FOR RELEASE

J. H. Foxgrover

J. H. FOXGROVER, RADM, USN
Commander, Naval Air Test Center

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INTRODUCTION

BACKGROUND

1. NAVAIRTESTCEN was directed by reference 1 to conduct flight tests to determine the single engine catapult launch capabilities of the EA-6B airplane. Tests were conducted to determine the single engine gross weight/temperature catapult launch envelope, the optimum piloting techniques and procedures to be utilized in the event of an engine failure during or immediately following catapult launch, and the feasibility of using 20 rather than 30 deg flaps for launches at heavy weights and high ambient temperatures. The results of these tests were reported in reference 2. In addition, tests were authorized to determine the EA-6B airplane's pitching moment coefficient in order to further study the single engine catapult launch environment through the use of computer simulation. This Technical Memorandum presents the normal and axial force, as well as the pitching moment and stability and control derivatives obtained from a maximum likelihood analysis of flight test data in the catapult launch configuration.

FLIGHT TEST DATA

2. The test data were obtained during two separate flights at an average altitude of 7,000 ft (2 130 m). One flight was in the clean (no stores) loading and one with three tactical jamming (TJ) pods and two 300-gal (1 135.6 ℓ) external fuel tanks installed. The test airplane utilized in this program, BuNo 156478, was aerodynamically representative of a production EA-6B airplane with the exception of the installation of a test nose boom containing an NACA pitot-static head, Kiel total pressure probe, and angle of attack and angle of sideslip vanes. The geometric and moment of inertia characteristics of the test airplane in the catapult launch configuration are presented in appendix A.

3. The evaluation was conducted in an aerodynamic configuration representative of the catapult launch environment (landing gear DOWN, flaps 30 deg, speedbrake IN). Thrust for level flight was utilized rather than the catapult launch setting of Military power in order to facilitate establishing trimmed level flight conditions. This did not have a significant effect on the estimated parameters. However, the test stabilizer trim requirements were slightly less than those which would be obtained at full Military power due to the reduced downwash characteristics at the horizontal stabilizer. The stabilizer required to trim with thrust for level flight is presented in figure 1.

1) NAVAIR msg 152018Z May 1975
2) NAVAIRTESTCEN Technical Report SA-105R-76, Evaluation of Single Engine Catapult Launch Capabilities of the EA-6B Airplane, Final Report, of 10 February 1977

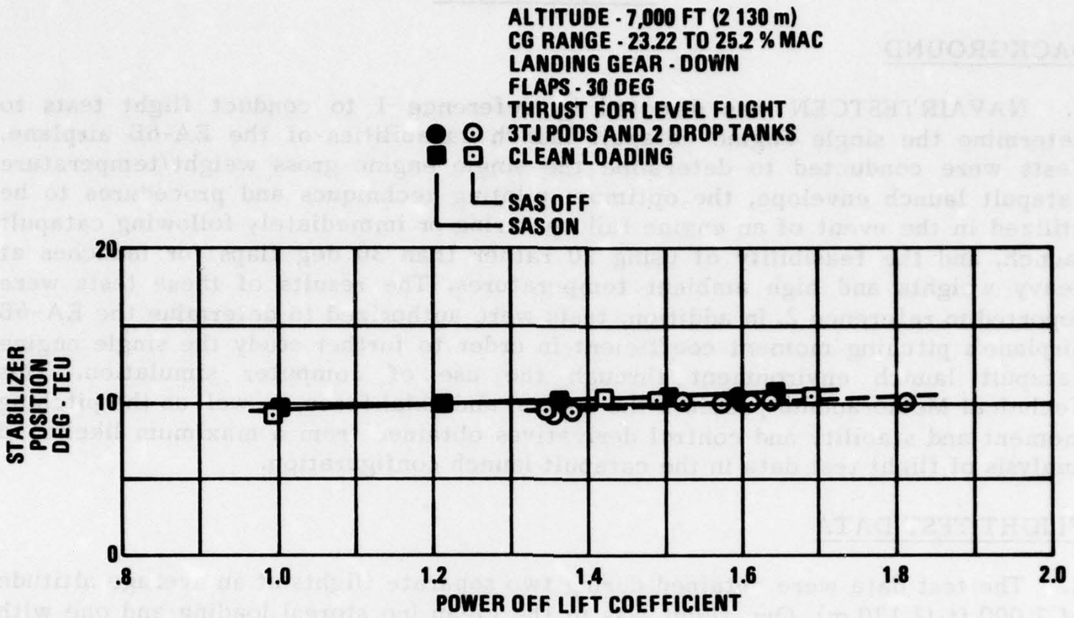


Figure 1
Level Flight Trim Requirements

4. The test airplane was instrumented for longitudinal flying qualities data as listed in appendix C. Test data were telemetered to the NAVAIRTESTCEN Real-Time Telemetry Processing System and recorded on onboard mangetic tape. The pilot techniques required to generate the flight test data necessary for the parameter identification process were standard dynamic longitudinal stability test techniques. The stability and control derivatives describing the short period mode of motion were obtained from the response to pitch doublets and sinusoidal stick pumping inputs. The stability derivatives describing the variation of forces and moments with respect to forward velocity were obtained from standard phugoid maneuvers. All tests were conducted from stabilized trim flight conditions.

PARAMETER IDENTIFICATION ALGORITHM

5. The parameter identification technique used in this analysis is a maximum likelihood identification algorithm (SCIDNT) developed in a joint program by NAVAIR, Office of Naval Research, NAVAIRTESTCEN, and Systems Control, Incorporated (SCI). Maximum likelihood estimation theory is discussed in references 3 and 4 and details of the computer algorithm used in this analysis are contained in reference 5. This maximum likelihood algorithm, as implemented, is basically a linear program capable of estimating linear airframe stability and control derivatives. The maximum likelihood identification procedures used provide for simultaneous estimation of aerodynamic coefficients and instrumentation errors. Thus, predata filtering or correction for instrumentation errors is not required prior to the parameter estimation procedure. The short period and phugoid response characteristics were analyzed at 25 and 5 samples per second, respectively.

6. The equations of motion (body fixed axis system) used in this analysis are presented in appendix B. The body axis system used is defined as follows:

a. Orthogonal, right-hand system, fixed to aircraft.

b. Origin fixed at aircraft cg.

c. X-axis is in the plane of symmetry fixed to fuselage reference line, positive forward.

d. Y-axis is perpendicular to the plane of symmetry, positive to the right (right wing).

e. Z-axis is in the plane of symmetry, perpendicular to the X-axis, positive downward.

7. The final set of aerodynamic coefficients presented in this report were determined by combining the results of numerous maneuvers at the same flight condition. The parameter estimates will vary between maneuvers because of the statistical nature of the random errors that occur in the measurement system and the different amounts of information in each maneuver concerning the parameters to be estimated (parameter identifiability). This variation in information or parameter identifiability is related to the degree of excitation of the modes of a system by a particular input and the corresponding ability to accurately identify the parameters of the system. In other words, in order to identify a mode (or parameters) of a system, that particular mode must be excited by the input.

3) NAVAIRTESTCEN Technical Report FT-77R-73, First Interim Report, Development of Digital Airframe Parameter Identification Technology, of 21 January 1974

4) NASA Contractor Report NASA CR-2200, Maximum Likelihood Identification and Optimal Input Design for Identifying Aircraft Stability and Control Derivatives, of March 1973

5) Systems Control, Incorporated, Engineering Report, SCIDNT I Theory and Applications - Technical Report No. 3, of December 1974

Parameter estimates will vary from run to run depending on the repeatability of the input or types of inputs used. Thus, as outlined in reference 6, the estimates from the different maneuvers must be combined to make full use of available data.

8. In terms of statistical characteristics, parameter estimate accuracy is discussed in terms of confidence bounds and estimated parameter error (EPE). Parameter estimate confidence bounds are determined in the identification algorithm based on the Cramer-Rao lower bound (references 4 and 5). A 1σ confidence bound represents a 67% confidence interval statement of the parameter estimates accuracy, and a 2σ confidence bound is a 95% confidence interval. For example, a 2σ confidence bound implies that, in a large number of flight tests under identical conditions, 95% of the time the parameter estimate will be within $\pm 2\sigma$ of the true parameter value. Parameter estimate confidence bounds are used as a guide in assessing the accuracy of individual parameter estimates; however, this confidence bound does not predict anything about the outcome of a single experiment in a definite sense. Thus, in addition to parameter estimate confidence bounds, other criteria must be considered, such as static stability, time history comparisons (response prediction criteria), and modal response frequency and damping calculations in assessing parameter estimate accuracy. The relative accuracy of all parameter estimates is qualitatively assessed in this report by using the EPE value which is defined as:

$$\text{EPE (Estimated Parameter Error)} = \frac{2\sigma}{\hat{p}} \times 100 \qquad (1)$$

where $\hat{p}$ is the parameter estimate and σ is the estimate of the 1σ confidence bound. Thus, EPE is the estimated parameter error in percent and is based on a 2σ confidence bound. Experimental results at NAVAIRTESTCEN indicate that the range of EPE values presented in table I can be used to express qualitatively the relative accuracy of parameter estimates.

Table I

Qualitative Assessment of Parameter Estimate Accuracy

EPE (%)	Qualitative Parameter Estimate Accuracy
0 - 9.0	Excellent
9.0 - 20.0	Good
20.0 - 36.6	Satisfactory
36.6 and above	Unacceptable

6) NAVAIRTESTCEN Technical Report SA-27R-76, Verification of S-3A Lateral-Directional Power Approach Characteristics Using a Maximum Likelihood Parameter Identification Technique, Final Report, of 28 May 1976

RESULTS AND DISCUSSION

DETERMINATION OF SHORT PERIOD RESPONSE CHARACTERISTICS

9. The aerodynamic coefficients describing the short period mode of motion at constant airspeed (Z_α, M_α, M_q, Z_{δ_e}, and M_{δ_e}) were determined from analysis of pitch doublet and sinusoidal stick input responses. Initially, attempts were made at including the variation of normal force with pitch rate (Z_q) in the estimation procedure. However, due to the small effect of this parameter on the total response, the estimated Z_q exhibited a very low confidence factor. Therefore, Z_q was held fixed at the wind tunnel value in all subsequent estimations. The other aerodynamic parameters (those that effect the phugoid mode) in the equations of motion were held at zero, since they did not affect the short period response.

10. The angle of attack response obtained from the initial estimations exhibited a lag of approximately 0.2 sec when compared to the measured angle of attack response. In order to account for this difference, the aircraft equations of motion were augmented with a description of the measurement lags in the noseboom system. This resulted in a sensed angle of attack described by the state equation:

$$\dot{\alpha}_s = -\left(\frac{1}{\tau\alpha}\right)\alpha_s + \left(\frac{1}{\tau\alpha}\right)\alpha_t - \left(\frac{1}{\tau\alpha}\right)\left(\frac{l_\alpha}{u_o}\right) q \qquad (2)$$

with the measured angle of attack being expressed as:

$$\alpha_m = K_\alpha \alpha_s + b_\alpha + \nu_\alpha \qquad (3)$$

In this manner, the lag time constant in the measured angle of attack ($1/\tau_\alpha$) was included as a parameter in the estimation procedure and resulted in better correlation between the estimated and measured angle of attack responses.

11. The nondimensional aerodynamic coefficients obtained from the maximum likelihood analysis of the short period responses are presented in appendix D, figures 1 through 3. The range of expected parameter errors resulting from the analysis of the individual maneuvers is presented in table II. The identification procedure yielded excellent accuracy in all of the parameters identified with the exception of Z_{δ_e} and $1/\tau_\alpha$ which were qualitatively assessed as good. A comparsion with the wind tunnel results of reference 7 shows excellent agreement in the normal force coefficients but large discrepancies in the pitching moment characteristics. The identified results indicate that the airplane is less statically stable and more dynamically stable than predicted by the wind tunnel data. In addition, the estimated M_{δ_e} shows that elevator deflections provide a significantly larger pitching moment than that estimated by the wind tunnel. Typical time history comparisons of the measured aircraft response, the maximum likelihood estimated response, and the wind tunnel predicted response are presented in appendix D, figures 4 and 5. The maximum likelihood estimated response compares favorably with the measured flight data while the wind tunnel predicted responses underestimate the magnitude of the response and exhibit less damping.

7) GAC Report XA1128-104-1, EA-6B Basic Stability and Control Data Report, of February 1968.

Table II

Short Period Parameter Estimate Accuracy

Parameter	Range of EPE (%)
M_{δ_e}	1 - 2
Z_{δ_e}	7 - 27
M_q	2 - 4
M_α	3 - 10
Z_α	1 - 4
K_α	2 - 5
$1/\tau_\alpha$	6 - 21

DETERMINATION OF LONG PERIOD RESPONSE

12. The additional derivatives required to describe the long period mode of motion (X_u, X_α, Z_u, M_u) were determined from analysis of phugoid responses. Difficulty was experienced in identifying the term X_α due to the lack of angle of attack response in the phugoid mode. However, since this term has a significant impact on the determination of phugoid damping and could not be ignored, it was calculated by the equation:

$$X_\alpha = \frac{\rho S\ V^2}{M}\left(C_L - C_{D_\alpha}\right) \tag{4}$$

and held fixed in the identification procedure. The other terms in the equations of motion, previously determined during the short period response analysis, were held fixed at their identified value during the long period identification procedure.

13. The phugoid data were analyzed over a time period beginning after the pilot's input was removed (returned to the trim position) and continuing through one and one-half to two cycles of aircraft motion. Thus, only the free response of the phugoid maneuver was used in the analysis (control derivatives X_{δ_e}, Z_{δ_e}, and M_{δ_e}, were set equal to zero). The initial force and moment conditions ($\dot{\bar{X}}_o$) were estimated at the beginning of each time period to take into accound the fact that the analysis was initiated from an off trim condition (see appendix B for a definition of $\dot{\bar{X}}_o$). The nondimensional aerodynamic coefficients obtained from the long period analysis are presented in table III. The identification procedure yielded excellent results in the estimation accuracy of C_{X_u} and C_{Z_u} and good results for C_{M_u}. A time history comparison of the measured flight data and the maximum likelihood estimated response, presented in appendix D, figure 6, shows excellent agreement in all of the measured responses.

Table III

Aerodynamic Coefficients Describing Variation in Force and Moment Characteristics with Forward Velocity

Loading - 3 TJ Pods and 2 Drop Tanks
True Angle of Attack - 14 deg
Landing Gear - DOWN
Flaps - 30 deg

Parameter	Value	EPE (%)
C_{X_u}	-.213	2.9
C_{Z_u}	-1.615	1.3
C_{M_u}	-.01	18.7

VERIFICATION OF IDENTIFIED PARAMETERS

14. The aerodynamic coefficients identified from the maximum likelihood estimation were verified through determination of the accuracy of the identified parameters (as based on estimated parameter error), time history comparisons of the measured and estimated responses, and comparison of computed response characteristics with conventional flight test results. The parameter accuracy (EPE) and time history comparisons made indicate that a high level of confidence should be placed on the identified parameters. A discussion of computed versus measured characteristics follows.

15. The frequency and damping characteristics of both the short and long period responses were obtained by first determining the Eigenvalues of the identified state matrix, F, and then calculating undamped natural frequency and damping ratio from the real and imaginary parts of those Eigenvalues. The resulting short period frequency and damping characteristics are presented in appendix D, figures 7 and 8. A comparison of the short and long period maximum likelihood, conventional flight test data analysis, and wind tunnel estimated frequency and damping characteristics is presented in table IV. Excellent agreement between the maximum likelihood estimated and conventional data analysis results are indicated at all test conditions. The wind tunnel estimates provide a good determination of the short period frequency characteristics but considerably underestimate the damping. This result is to be expected from the much lower pitch damping term ($C_{M_q} + C_{M_{\dot{\alpha}}}$) obtained from the wind tunnel data (see paragraph 11).

Table IV

Frequency and Damping Characteristics
Loading - 3 TJ Pods and 2 Drop Tanks
Landing Gear - DOWN
Flaps - 30 deg

True Angle of Attack (deg)	Modal Parameter		Maximum Likelihood Identification Result	Conventional Data Analysis Result	Wind Tunnel Estimate
12	Short Period	$\omega_{n_{SP}}$ (rad/sec)	1.05	1.0	1.05
		ζ_{SP}	0.68	0.61	0.45
18.5	Short Period	$\omega_{n_{SP}}$ (rad/sec)	0.88	1.0	1.04
		ζ_{SP}	0.67	0.66	0.34
14	Long Period	ω_{n_p} (rad/sec)	0.148	0.147	
		ζ_p	0.092	0.093	-

16. Additional verification of the maximum likelihood estimated parameters was obtained by comparison with conventional static longitudinal stability test results. Static longitudinal stability characteristics in terms of elevator position gradients with airspeed can be determined by evaluating the forward airspeed to horizontal stabilizer transfer function at steady state conditions:

$$\frac{u(s)}{\delta_e(s)} = \frac{N^u_{\delta_e}}{\Delta} = \frac{K_u\left(s + 1/\tau_1\right)\left(s + 1/\tau_2\right)}{\left(s^2 + 2\zeta_p\,\omega_{n_p}\,s + \omega_{n_p}^2\right)\left(s^2 + 2\zeta_{SP}\,\omega_{n_{SP}}\,s + \omega_{n_{SP}}^2\right)} \tag{5}$$

Evaluating this equation at s = o (steady state) yields

$$\left.\frac{u(s)}{\delta_{e(s)}}\right|_{s=o} = \frac{K_u\left(1/\tau_1\right)\left(1/\tau_2\right)}{\omega_{n_p}^2\,\omega_{n_{SP}}^2} \tag{6}$$

which is equivalent to

$$\frac{u(s)}{\delta_{e(s)}}\bigg|_{s=o} = \frac{-\left(-Z_\alpha \; M_{\delta_e} + M_\alpha \; Z_{\delta_e}\right)}{\left(M_\alpha \; Z_u - M_u \; Z_\alpha\right)} \tag{7}$$

in terms of the identified parameters. Equation (7) can be interpreted as the gradient of airspeed with elevator position during a static longitudinal stability test. The static longitudinal stability results utilizing the estimated parameters obtained from dynamic stability tests are compared with the conventional results of static longitudinal tests (reference 8) in appendix D, figure 9. Again, excellent agreement is obtained between the two methods, thereby lending further credence to the parameters identified by the maximum likelihood algorithm.

8) NAVAIRTESTCEN Technical Report FT-68R-73, Flying Qualities and Performance Technical Evaluation of the EA-6B Airplane, Final Report, of 30 October 1973

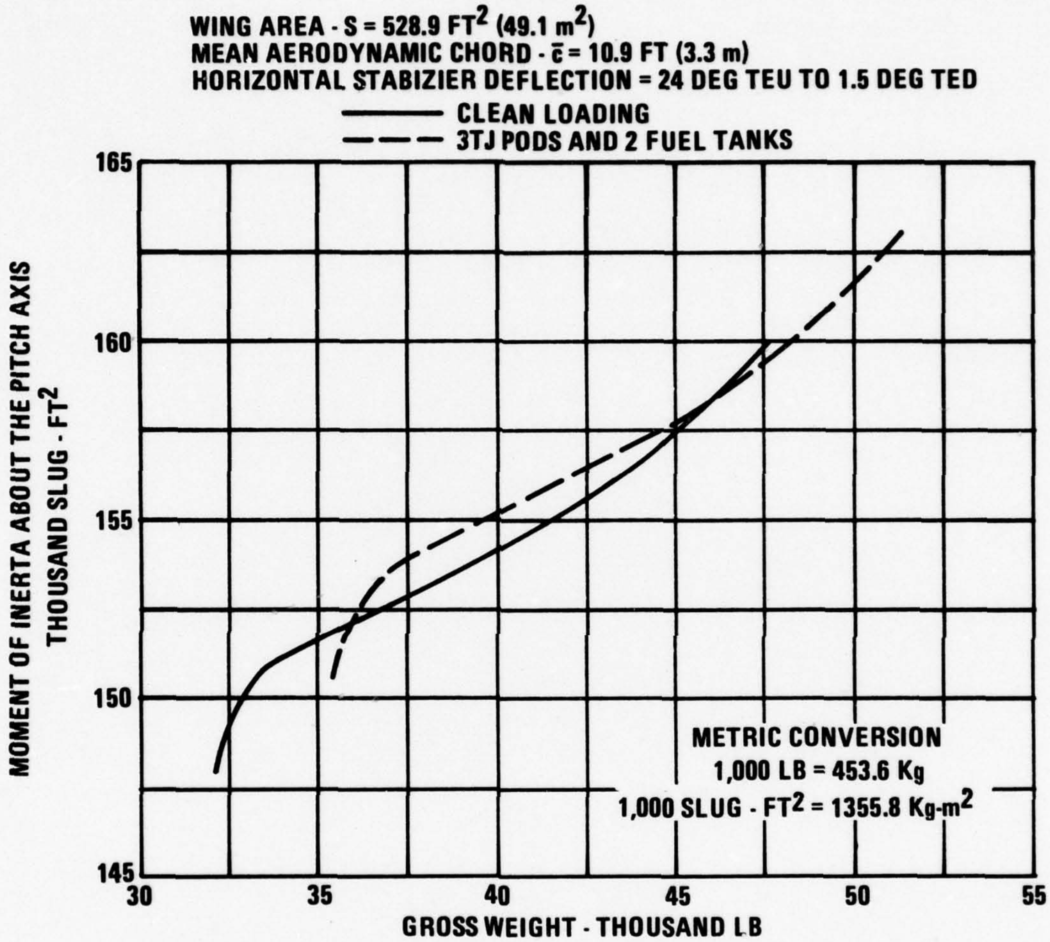
CONCLUDING REMARKS

17. The stability and control derivatives describing the longitudinal aerodynamics of the EA-6B airplane in the catapult launch configuration have been accurately determined from flight data through the use of a maximum likelihood estimation algorithm. The identified derivatives were obtained from a trimmed level flight condition and are useful for point simulations and specification determination. If the simulation of out of trim flight conditions is required, tests would have to be conducted at other than level flight conditions in order to determine the effects of thrust and horizontal stabilizer trim settings on the aerodynamic parameters.

18. The aerodynamic coefficients were generated in a form suitable for the construction of a data base for simulation of EA-6B airplane characteristics. This data base has direct application to a parametric study of catapult launch capabilities of the EA-6B airplane through computer simulations. The data base can also be used for optimization of control system configurations in ACLS simulations, for verifying or constructing the data base of the EA-6B Operational Flight Trainer, and for determining aircraft modal parameters for specification analysis.

GEOMETRIC CHARACTERISTICS
Landing Gear and Flaps Extended

EA-6B Airplane
BuNo 156487



EQUATIONS OF MOTION

LINEAR STATE SPACE MODEL

1. The longitudinal equations of motion and measurement equations used in the maximum likelihood parameter identification algorithm were written in standard linear state space format in a body fixed axis system.

Equation of Motion:

$$\dot{\bar{x}} = F\bar{x} + G\bar{u} + \dot{X}_o$$

where:

$\bar{x}$ is an nxl state vector

$\bar{u}$ is a pxl control vector

F is a nxn matrix of stability derivatives

G is a nxp matrix of control derivatives

$\dot{X}_o$ is an nx1 vector of initial conditions on the state vector

Measurement Equation:

$$\bar{y} = H\bar{x} + D\bar{u} + \bar{b} + \bar{\nu}$$

where:

y is an rx1 measurement vector

b is an rx1 measurement bias vector

$\bar{\nu}$ is an rx1 measurement error vector

H is an rxn matrix relating the measurement to the state vector

and:

n is the number of states

p is the number of controls

r is the number of measurements

This type of formulation is very general and allows for the maximum amount of flexibility in including the correct relationship between the dynamic equations of motion and measurements of aircraft response variables. Thus, errors in instrumentation system measurements are included as an integral part of the identification

process (measurement errors are estimated simultaneously with aircraft stability and control derivatives). For example, during the identification process, angle of attack measurements are corrected for pitch rate effects on the local flow field at the nose boom and instrument bias, measurement random error, and angle of attack vane scale factor are estimated.

LONGITUDINAL MODEL

2. For the longitudinal case, the state space model takes on the following form:

Equations of Motion:

State vector: $\bar{x} = \left[\alpha, u, q, \theta, \alpha_s\right]^T$

where α_s, the sensed angle of attack was included to account for lags in the angle of attack sensing system of the test airplane.

Control vector: $\bar{u} = \left[\delta_e, 1\right]^T$

$$F = \begin{bmatrix} Z_\alpha & Z_u & (Z_q + 1) & \frac{-g \sin \theta_o}{V_o} & 0 \\ X_\alpha & X_u & (X_q - \alpha_o \ V_o) & -g \cos \theta_o & 0 \\ M_\alpha & M_u & M_q & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ \frac{1}{\tau_\alpha} & 0 & -\left(\frac{1}{\tau_\alpha}\right)\frac{\alpha}{V_o} & 0 & -\frac{1}{\tau_\alpha} \end{bmatrix}$$

$$G = \begin{bmatrix} Z_{\delta_e} & Z_o \\ X_{\delta_e} & X_o \\ M_{\delta_e} & M_o \\ 0 & q_o \\ 0 & \alpha_{s_o} \end{bmatrix}$$

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TEST INSTRUMENTATION PARAMETERS
EA-6B Airplane
BuNo 156478

Parameter	Cockpit	Mag Tape and TM
Nose Boom Airspeed		X
Nose Boom Altitude		X
Nose Boom Angle of Attack		X
Nose Boom Sideslip Angle		X
Production Airspeed	X	X
Production Altitude	X	X
Production Angle of Attack	X	X
Outside Air Temperature	X	X
Left and Right Turbine Discharge Pressure		X
Left and Right Low Speed Compressor RPM		X
Left and Right High Speed Compressor RPM		X
Left and Right Turbine Outlet Temperature		X
Left and Right Fuel Used Counters (Volume)		X
Horizontal Stabilizer Position		X
Pitch Angle		X
Longitudinal Stick Position		X
Pitch Rate		X
CG Normal Acceleration		X
Flaperon Position		X
Time Code		X

DATA

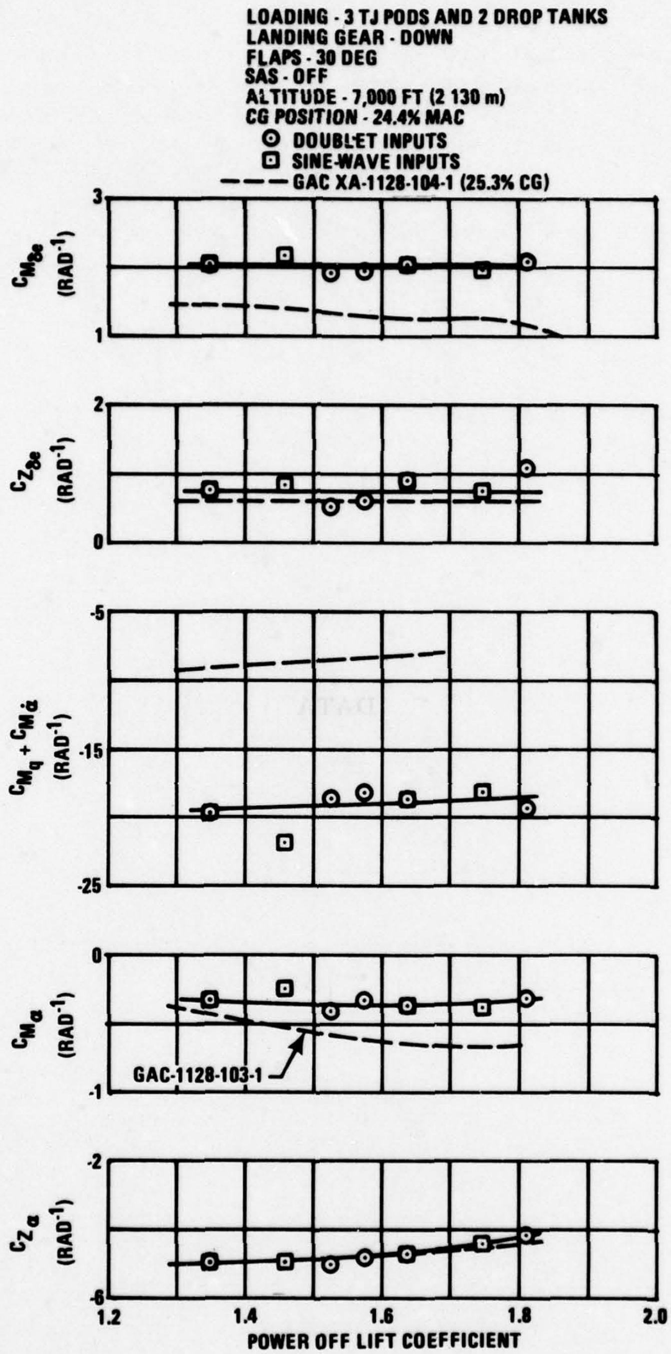

Figure 1
EA-6B Airplane
BuNo 156478
Short Period Aerodynamic Coefficients

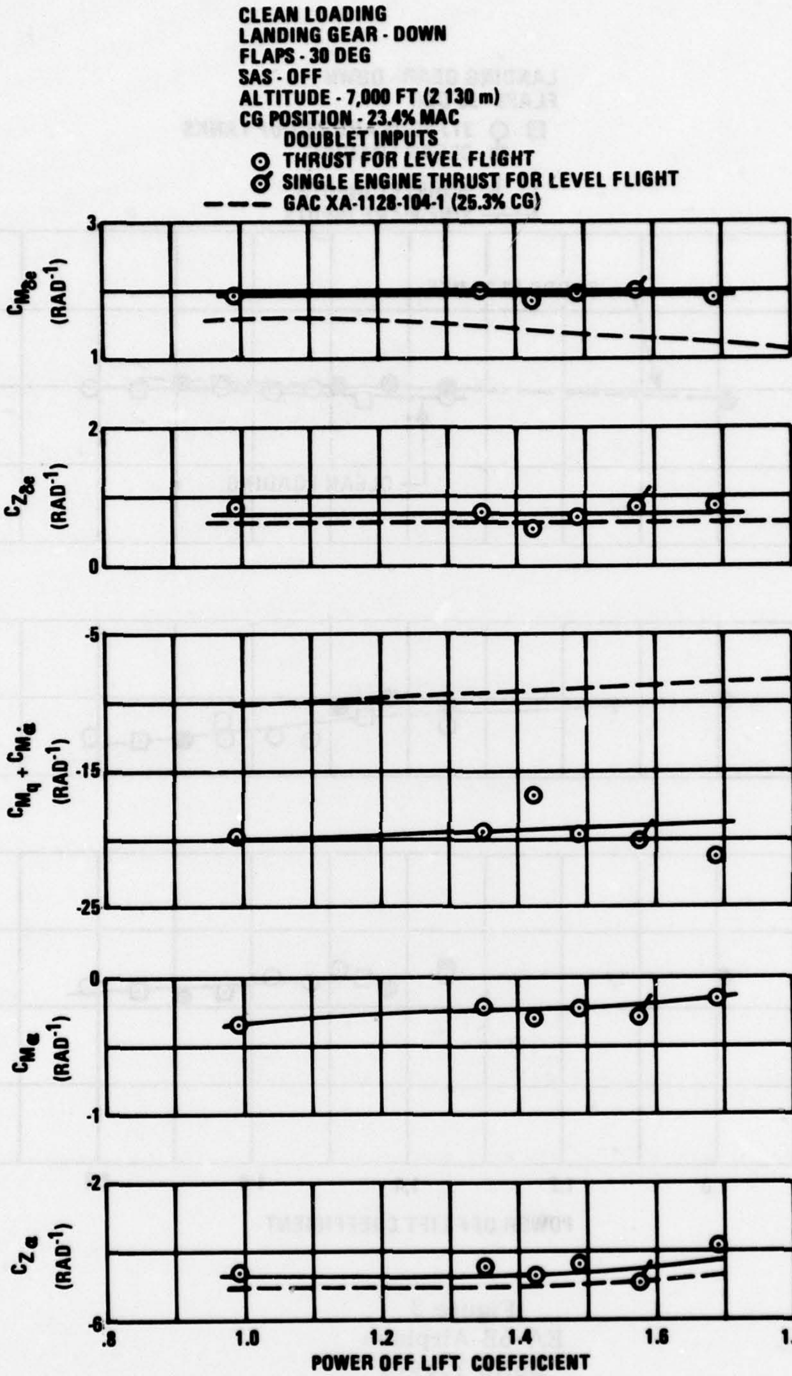

Figure 2
EA-6B Airplane
BuNo 156478
Short Period Aerodynamic Coefficients

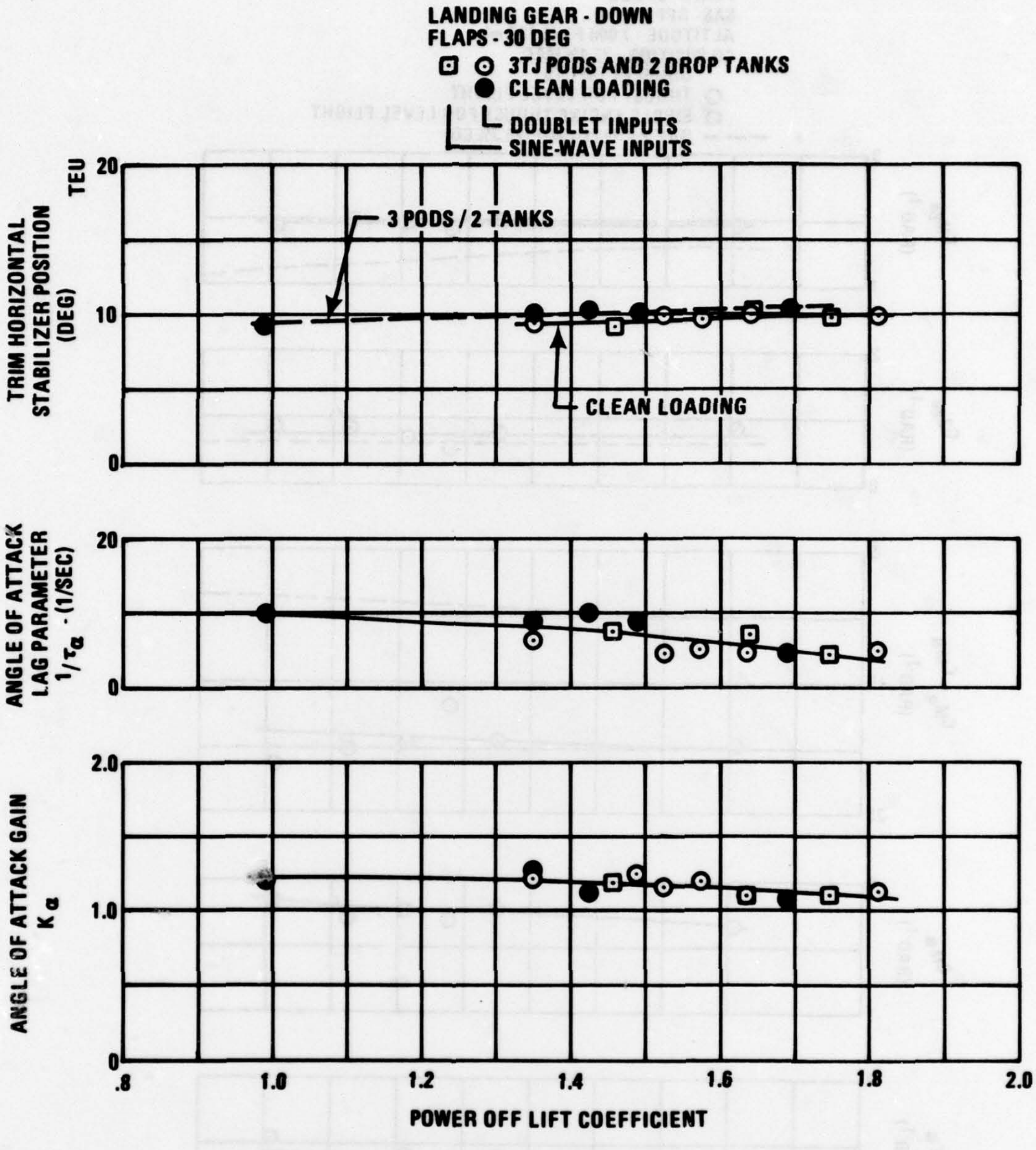

Figure 3
EA-6B Airplane
BuNo 156478
Angle of Attack and Stabilizer Characteristics

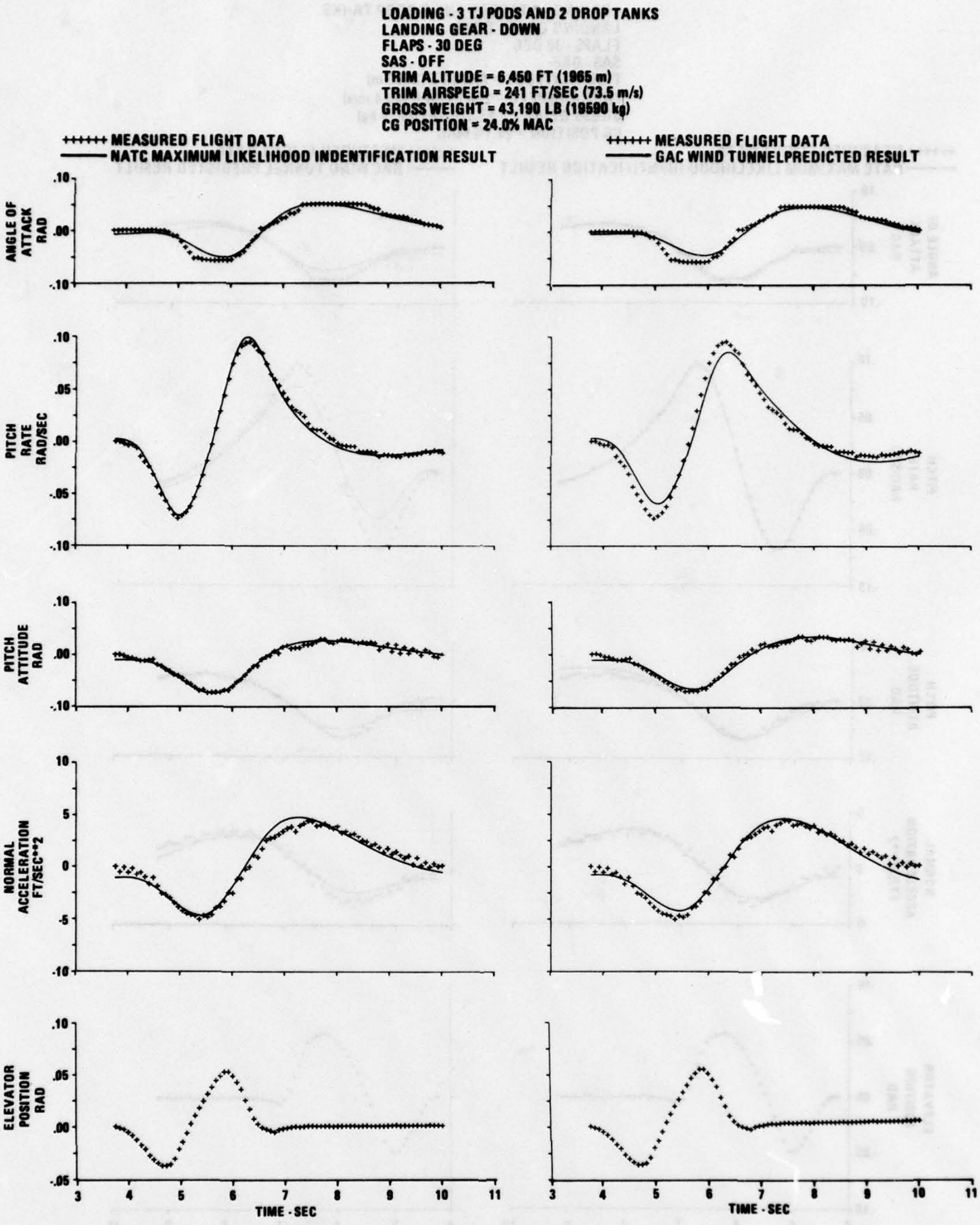

Figure 4
EA-6B Airplane
BuNo 156478
Time History Comparison of Short Period Characteristics

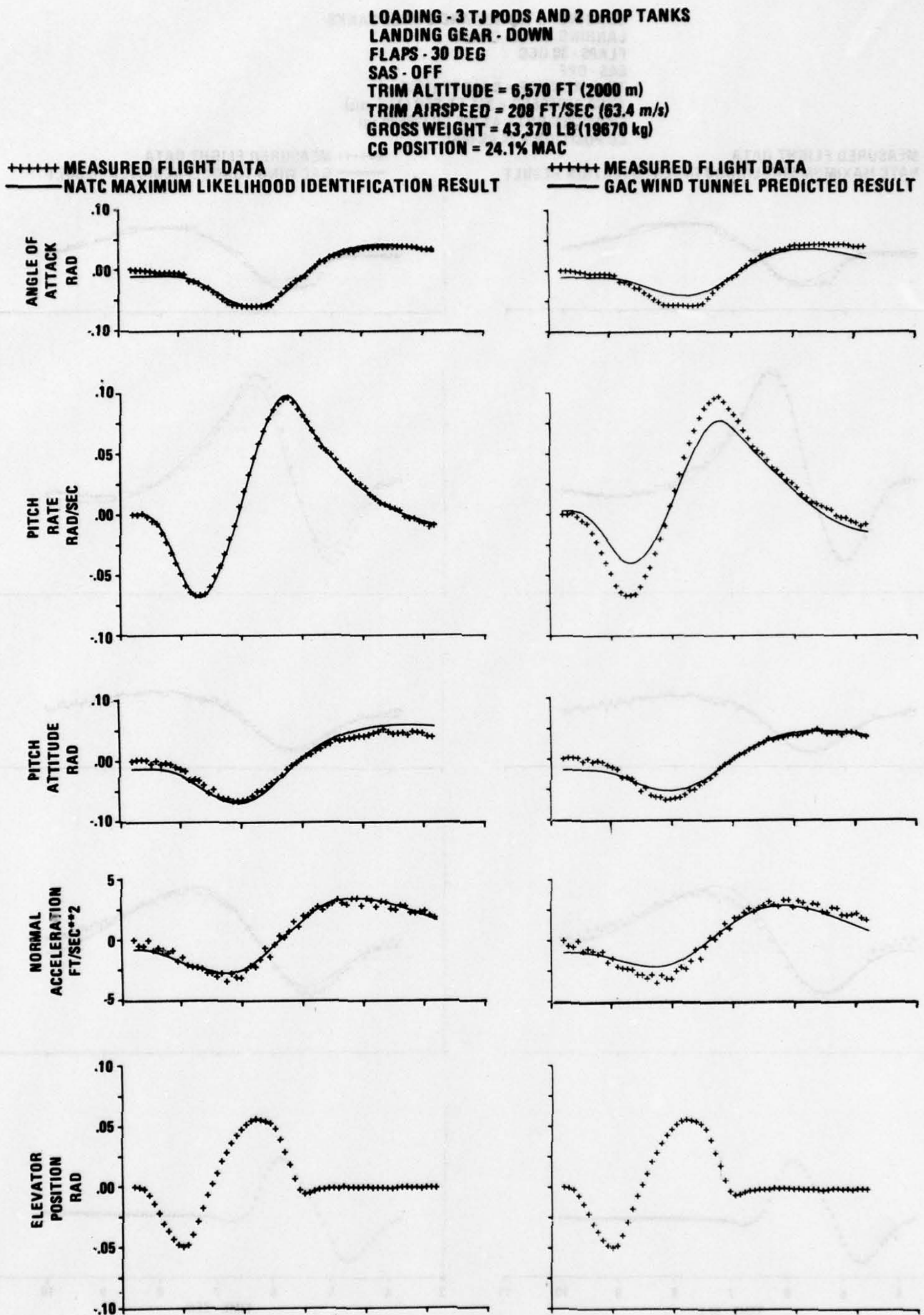

Figure 5
EA-6B Airplane
BuNo 156478
Time History Comparison of Short Period Characteristics

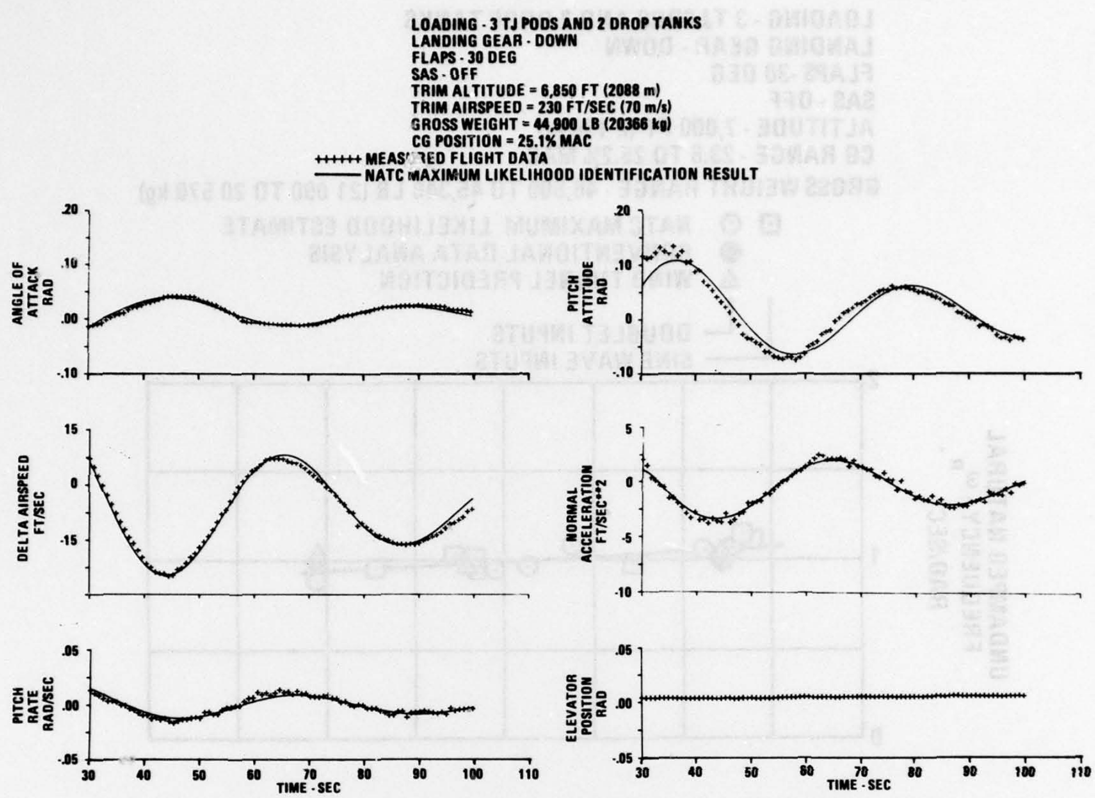

Figure 6
EA-6B Airplane
BuNo 156478
Time History Comparison of Long Period Characteristics

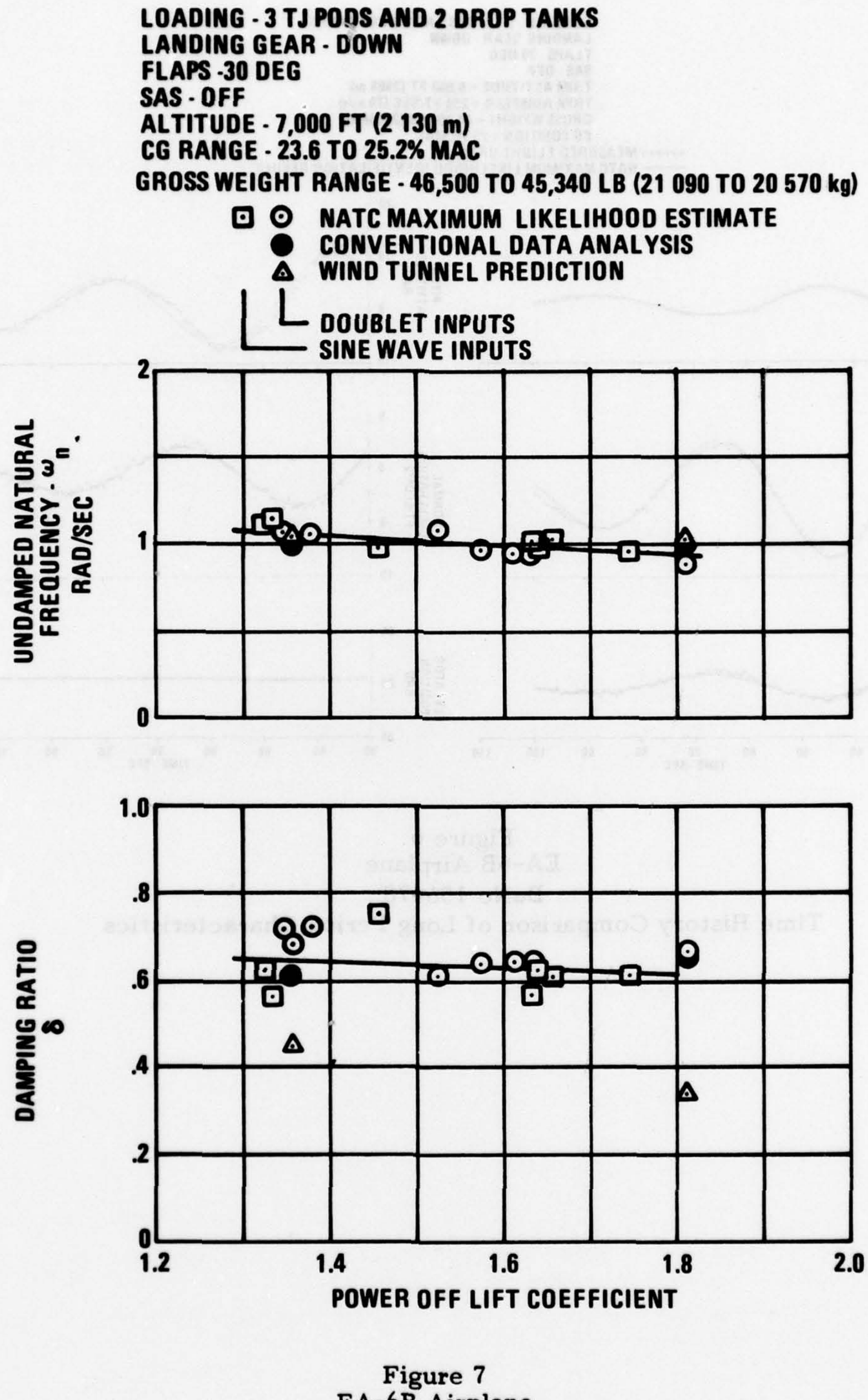

Figure 7
EA-6B Airplane
BuNo 156478
Short Period Frequency and Damping Characteristics

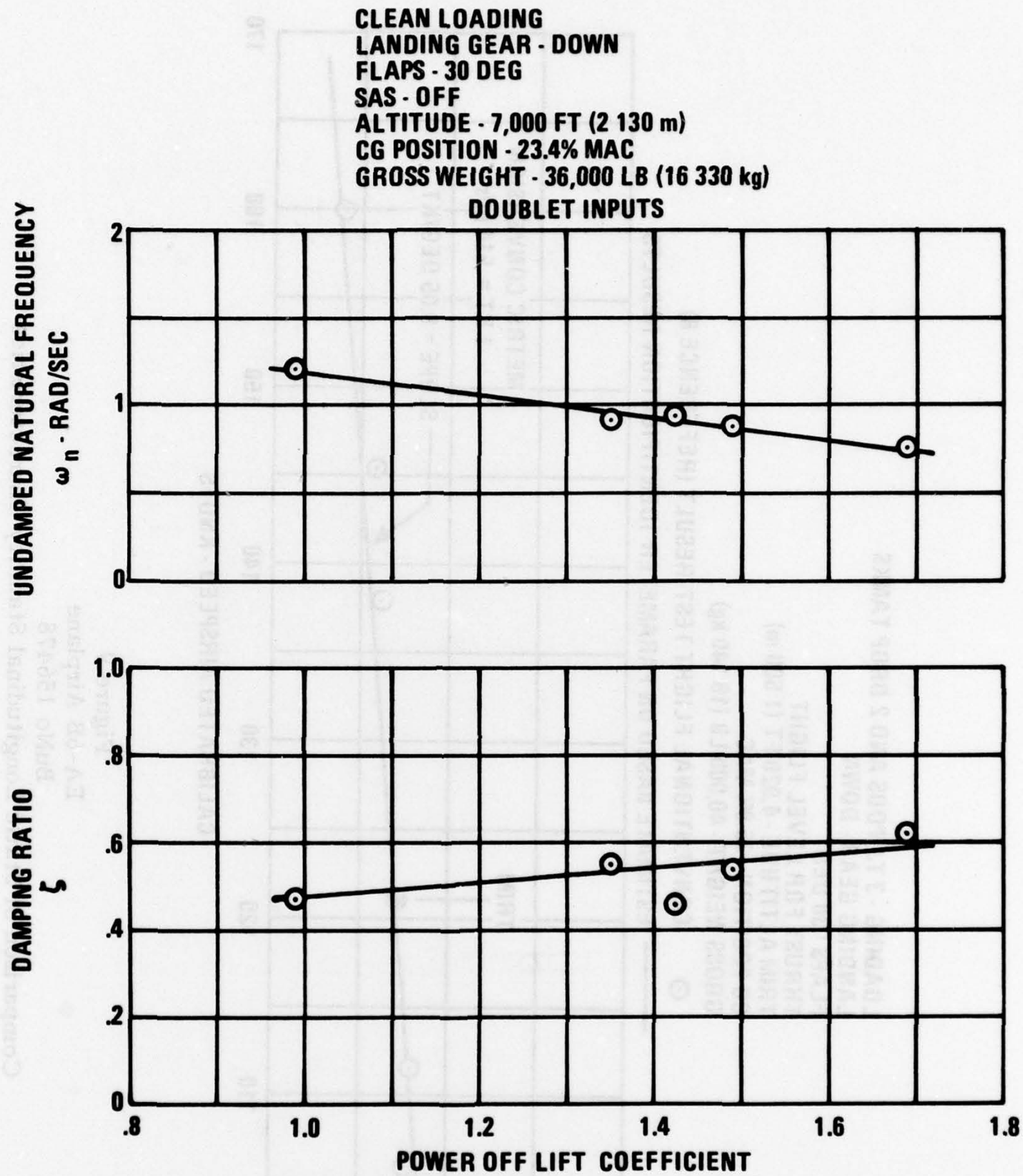

Figure 8
EA-6B Airplane
Short Period Frequency and Damping Characteristics

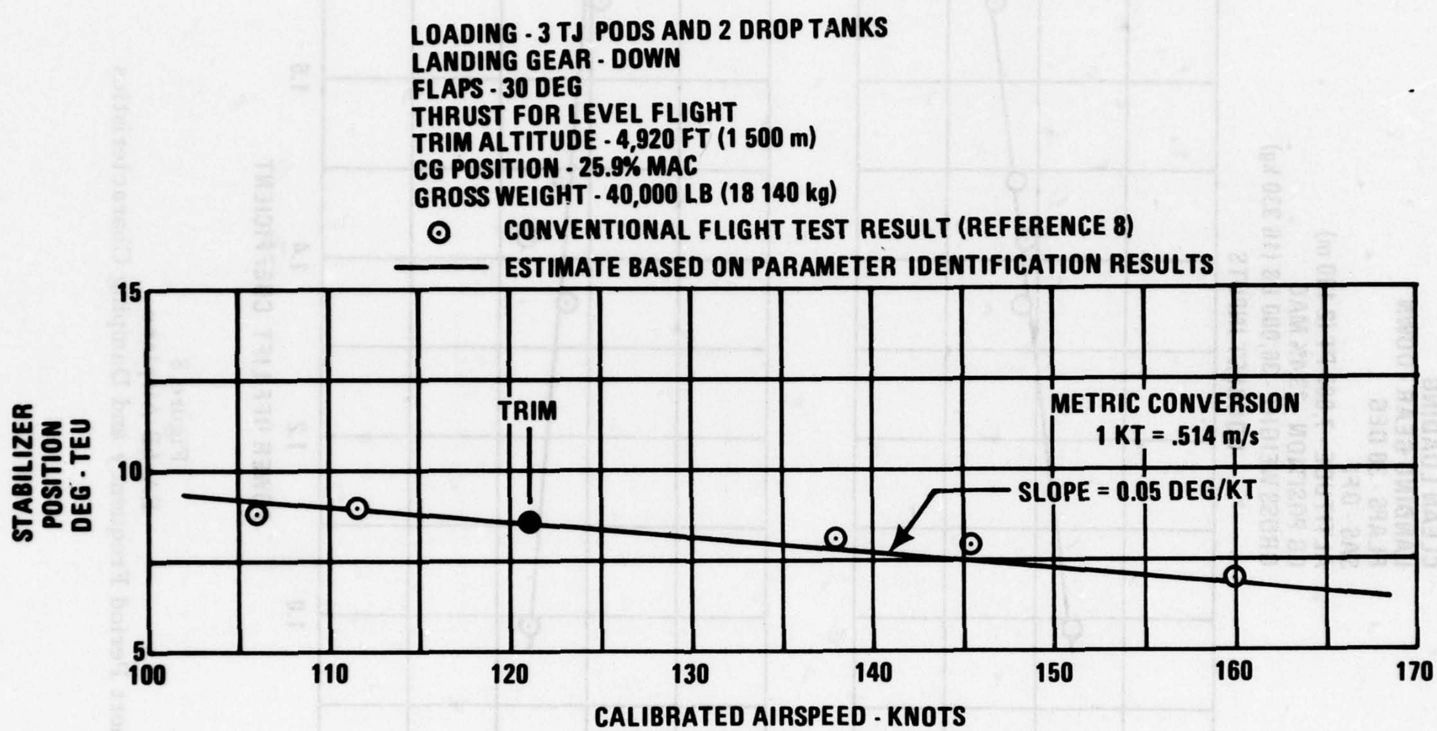

Figure 9
EA-6B Airplane
BuNo 156478
Comparison of Static Longitudinal Stability Characteristics

LIST OF SYMBOLS

Symbol	Definition	Units	
a_z	Vertical acceleration	ft/sec^2	(m/s^2)
b	Measurement bias		
$\bar{c}$	Mean aerodynamic chord	ft	(m)
C_{K_x}, C_{K_δ}	Nondimensional partial derivative of K force or moment (X, Z, M) with respect to state variable, x, or control, δ	-	-
$C_{L_{PWR\ OFF}}$	*Power-off lift coefficient*	-	-
D	Matrix relating measurements to control vector		
EPE	Estimated Parameter Error	%	
F	Matrix of stability derivatives		
G	Matrix of control derivatives		
g	Acceleration due to gravity	ft/sec^2	(m/s^2)
H	Matrix relating measurements to state vector		
I_{YY}	Moment of inertia about pitch axis	slug-ft^2	(kg-m^2)
$K\alpha$	Angle of attack vane scale factor		
l_x	X distance of angle of attack vane to cg	ft	(m)
l_z	Z distance of angle of attack vane to cg	ft	(m)
M	Pitching moment about Y axis	ft-lb	(N-m)
M_x, M_δ	Partial derivative of pitching moment with respect to state variable, x, or control, δ		
$\hat{p}$	Parameter estimate (general)		
q	Pitch rate	rad/sec	

Symbol	Definition	Units	
$\bar{q}$	Dynamic pressure	lb/ft^2	(Pa)
s	Laplace operator	-	
S	Wing area	ft^2	(m^2)
t	Time	sec	
u	Longitudinal component of velocity	ft/sec	(m/s)
V	Free stream velocity	ft/sec	(m/s)
w	Vertical component of velocity	ft/sec	(m/s)
x	State variable (α , u, q, θ, α_s)	-	
X	X component of force	lb	(N)
X_x, X_δ	Partial derivative of X component of force with respect to state variable, x, or control, δ		
y	Measurement variable (α_m, u_m, q_m, θ_m, a_{z_m})		
Z	Z component of force	lb	(N)
Z_x, Z_δ	Partial derivative of Z component of force with respect to state variable, x, or control, δ		
$(\dot{\ })$	Time rate of change	sec^{-1}	
$(\hat{\ })$	Estimate		
$(\bar{\ })$	Vector	rad	
α	Angle of attack		
δ	Control deflection	rad	
i_s	Horizontal stabilizer position	rad, deg	
ζ	Damping ratio	-	
θ	Pitch attitude	rad	
σ	Parameter estimate variance or confidence bound	-	
τ	Time constant	sec	

Symbol	Definition	Units
ν	Measurement random error	
ω_η	Natrual frequency	rad/sec

Subscripts

m	Measured	
t	True	
$( )_o$	Trim condition	

Superscripts

-1	Inverse	
T	Transpose	

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